REPORT NUMBER 148
MARCH 1964

INSTALLED SYSTEMS FUNCTIONAL TEST SUMMARY

XV-5A

LIFT FAN FLIGHT RESEARCH AIRCRAFT PROGRAM

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GENERAL ELECTRIC

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REPORT NUMBER 148
MARCH, 1964

INSTALLED SYSTEMS FUNCTIONAL TEST

SUMMARY

XV-5A LIFT FAN

FLIGHT RESEARCH AIRCRAFT PROGRAM

ADVANCED ENGINE AND TECHNOLOGY DEPARTMENT

GENERAL ELECTRIC COMPANY

CINCINNATI, OHIO 45215

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1.0 INTRODUCTION

This report presents a summary of the installed systems functional tests performed on XV-5A aircraft S/N 24506, by the Ryan Aeronautical Company at Lindbergh Field, San Diego, during the period from 21 November 1963 to 25 February 1964.

Unless specified to the contrary in the text of this report, the test procedures followed were those presented in Report No. 136 "XV-5A Installed Systems Functional Test Procedure" dated November, 1963.

XV-5A aircraft S/N 24506 is the first aircraft scheduled to enter the flight test program at Edwards Air Force Base. Certain tests which were conducted using XV-5A aircraft S/N 24505 are also applicable to S/N 24506 and are so designated.

Presentation sequence in this report follows that of Report No. 136 for ease of comparison.

In cases where systems required modification as a result of tests, the tests were repeated and the results shown herein are for the final configuration.

This summary report should be considered as an interim report only and will be followed by the final data report now in preparation.

All tests were conducted under the surveillance of Ryan Quality Control and the XV-5A Design Team. It is Ryan's conclusion that the XV-5A aircraft S/N 24506 has been ground tested in all areas pertaining to hover, forward fan supported flight, and low speed conventional flight and is acceptable to proceed into active flight testing.

2.0 TEST RESULTS

2.1 ELECTRICAL SYSTEM CHECKOUT

The aircraft was tested as described in Section 3.1 of Report No. 136. All systems operated per their design requirements and the electrical system was cleared for flight.

During ground tests the need for several minor electrical system changes were identified:

- Removal of the horizontal stability trim switches which are activated with stick displacement during hovering flight.
- Relocating the emergency horizontal stabilizer trim relay to a location close to the stabilizer actuator.

These items will be incorporated at Edwards Air Force Base prior to transition flight and will be tested as a supplementary test of Section 3.1 of Report No. 136.

2.2 SURFACE GAINS AND HYSTERESIS

Surface gain and hysteresis tests as specified in Report No. 136 were performed. The following items were noted from the recorded data:

1. At maximum collective stick position, the LH Aft louver mixer mechanism encountered interference for a right wing down operation.
2. At minimum collective stick position, the RH Aft louver mixer mechanism encountered interference for right wing down operation.
3. Full rudder travel and full louver travel were not available for right yaw input from the rudder pedals.
4. The force required to move the collective stick in the up direction is approximately twice that required for down direction.

The above discrepancies were corrected and retested. The results of all control surface gains and hysteresis tests were considered satisfactory.

2.3 FLIGHT CONTROLS STABILITY

These tests were performed on aircraft S/N 24505 in accordance with the referenced test procedure section of Report No. 136 and the deviations stated below.

2.3.1 Lateral Stick to Aileron - CTOL Mode

The vibration was maintained at a double amplitude displacement of .20 inches from .5 cps to 30 cps and the response curves were obtained.

As a result of preliminary tests, the aileron servo control valves were modified from an underlapped to an overlapped configuration and the longitudinal push rods to the aileron servos were stiffened. Retests of the lateral stick to aileron-CTOL mode met all design requirements.

2.3.2 Lateral Stick to Wing Fan Louver Servo

The over-all vibration response was determined by combining the responses from the lateral stick to the aileron droop mechanism crank input, and from the aileron droop mechanism crank input to the louver servos. This two-step operation was required to avoid excessive control acceleration.

As a result of initial tests, the cockpit floor was reinforced in the vicinity of the stick and the pivot bracket for the 143C045-9 linkage at Station 156.6 was reinforced. This test was repeated with the stability characteristics meeting design requirements.

2.3.3 Longitudinal Stick to Pitch Fan Door Servo

The vibration at the stick was maintained at a double amplitude displacement of .22 inches. The damping ratio found at 9 cps was approximately .0562 and at 30 cps was approximately .0281. These quantities were calculated from the frequency response curves. Although the damping ratio at 30 cps is somewhat less than anticipated, no problem will occur because there is positive dampening.

2.3.4 Rudder Pedal to Wing Fan Louver Servo

The rudder control cables were disconnected aft of the forward cable tension regulator. The rudder pedals were removed and the vibration was applied to the pedal torque tube in the cockpit at a double amplitude of .24 inches. This procedure was followed to avoid damage to the rudder and to the rudder pedals caused by excessive controls accelerations.

Initial testing damaged the rudder pedals, which were replaced, and made it necessary to modify the yaw crank input to the mechanical mixer to a double bearing configuration. This test met all of the design objectives after completion of the modification.

2.3.5 Collective Control to Wing Fan Louver Servo

The vibration at the collective control was maintained at a double amplitude of 1.0 inch. This test indicated that the stability characteristics meet the design requirements.

2.3.6 Step Inputs

The transient (step input) tests were conducted as outlined in Report No. 136, Paragraph 3.3.6, except that a Sanborn Model 60-1300 recorder was used.

1. Longitudinal stick to elevator CTOL mode.
    a. Natural frequency 8.13 cps down elevator, and 9.53 up elevator direction.
    b. Damping ratio .0748.
2. Longitudinal stick to pitch control door servo.
    a. Natural frequency 8.0 cps. Down elevator, and 10.0 cps up elevator direction.
    b. Damping ratio .070.
3. Rudder pedal to rudder CTOL mode.
    a. Natural frequency 8.19 cps right rudder, and 7.69 cps left rudder direction.

b. Damping ratio .0459.

4. Rudder pedal to L/H aft louver servo.

a. Natural frequency 8.70 cps.

b. Damping ratio .0556.

These tests indicate that the longitudinal and lateral control in the VTOL flight are adequate. Additional studies will be conducted with regard to the aircraft stability during VTOL in the areas of control transients, and gust load conditions.

2.4 FLIGHT MODE CONVERSION SEQUENCE

The pitch fan inlet louver position was not recorded because the louvers had been removed for rework when the test was conducted. However, the pitch fan inlet louver position was recorded on the XV-5A aircraft S/N 24505 and successfully met the design objectives. The diverter valve transient time was determined while converting from VTOL to CTOL position only on the XV-5A aircraft S/N 24506. The diverter valve transient time was determined both ways on XV-5A aircraft S/N 24505.

These tests were also conducted during engine runs at 70% rpm on both engines and on primary and standby electrical power. The tests were repeated at 90% rpm on both engines on primary electrical power only.

Conversion abort tests were also successfully conducted at 70% rpm on both engines.

2.5 COCKPIT CHECKOUT

The cockpit checks were conducted to demonstrate the operation and adequacy of the pitot-static system, throttle, canopy lock, and spin chute installations. All systems performed per design requirements.

2.5.1 Pitot-Static System

All pressure sensing switches and instruments were removed from the pitot and static lines and both lines were checked for flow, and subsequently pressure tested to 20 psig.

The landing gear warning pressure switches were placed in the system and were found to operate as prescribed.

The pilot's seat speed sensor arming switch was tested to manufacturer's specifications in the laboratory and an operational check was made. All functions were normal. Pressure sensing instruments were calibrated in the laboratory and checked for normal operation in the aircraft. The ejection seat system will be tested again at the time pyrotechnics are installed before the first flight.

Tests of the VTOL mode air-speed system resulted in slight changes to that system. A calibrated bleed and a restrictor were added in order that sudden changes in pitot pressure, at the time of mode change, would not overdrive the air-speed indicator. The resulting system produces a one-to-two knot drop in indicated air-speed during VTOL operation.

2.5.2 Power Quadrant and Engine Controls

Both engine throttle systems were checked individually and collectively. Slight discrepancies in throttle lever friction clutching were noted but may be corrected through proper adjustment of the friction mechanism. Twist grip torque was within the limits specified by design. Twist grip rotation and resulting throttle lever displacement were as specified.

2.5.3 Canopy Latch

Canopy down lock and intermediate stops were checked for compliance with design requirements. The down lock lever secured the canopy against the canopy seals with shear pins engaging the receivers. The down lock was positive with only moderate latch lever forces required.

2.5.4 Drag Anti-Spin Chute

Deployment of the spin chute was checked and found to be satisfactory. Jettisoning procedures were tested with no discrepancies noted.

2.5.5 Automatic Throttle Cutback

Automatic throttle cut-backs resulting from fan overspeed were tested. Overspeed signals were simulated using General Electric test equipment. Throttles were in the 90% rpm position and a conversion from CTOL to VTOL flight mode was made. As the wing fan louvers passed through 45°, the pointers on the engine fuel control boxes cut back to approximately

the 70% rpm position. The throttle cutback switch reset the pointers to the original 90% rpm position.

The system was operationally checked during engine run test, reference paragraph 2.6.13 of this report.

2.6 ENGINE RUN TEMPERATURE SURVEY

This series of tests was run as described in the following paragraphs. On occasions, local over-temperature conditions were experienced and modifications were incorporated to remedy the situation. These modifications are discussed in the applicable paragraphs. Results were satisfactory unless otherwise noted.

2.6.1 Ignition Test

The ignition switch was turned on (for each engine individually) for approximately 10 seconds. Both ignitors were audible.

2.6.2 Motoring Test

Each engine was turned over several revolutions by hand. No rubbing, dragging or unusual sounds were noted. Each engine was then motored with the air-start cart at 15% rpm and fuel flow checked. No problems were encountered. Engine run-down times were logged for each subsequent engine operation.

2.6.3 Idle - CTOL

The engines were run at idle rpm for approximately five minutes individually in the CTOL mode. A leak check and general inspection was performed during this run.

2.6.4 Single Engine Power Run - CTOL

The engines were run individually in the CTOL mode to 98% rpm in the following increments: 48%, 70%, 80%, 90%, 95% and 98%. Higher rpm's were not obtainable because of a throttle adjustment. Each power setting was held until the EGT stabilized (approximately 10 seconds).

2.6.5 Single Engine Idle - VTOL

The engines were run individually at idle for approximately five minutes in the VTOL mode at $\beta_V = 0$.

2.6.6 Single Engine Power Run - VTOL

The engines were run individually in the VTOL mode to 98% rpm in the following increments: 70%, 80%, 90%, 95% and 98%. Higher rpm's were not obtainable due to improper throttle adjustment. Each power setting was held until the EGT stabilized.

2.6.7 Dual Engine Power Run - CTOL

The engines were run simultaneously in the CTOL mode to 100% in the following increments: 48%, 70%, 80%, 90%, 95%, 98%, 99%, and 100%. Each power setting was held for approximately three minutes up to 100% rpm which was held for five minutes for heating criteria compliance.

The lower fuselage skin just aft of the tailpipe openings was overheated during dual engine runs. The damaged skin was replaced and the existing external insulation was extended to cover the area.

Chine rails were also installed on both sides of the aft fuselage to prevent hot gases from "washing" the upper aft fuselage.

2.6.8 Dual Engine VTOL Power Runs

Power Setting (%)	Mode	β_V	Time (Minutes)
70	VTOL	0°	2
70	CTOL		2
70	VTOL	-5°	2
70	CTOL		2
70	VTOL	20°	2
70	CTOL		2
70	VTOL	40°	2
70	CTOL		2
90	VTOL	0°	2
70	CTOL		2
90	VTOL	20°	2
70	CTOL		2
90	VTOL	30°	2
70	CTOL		2
90	VTOL	40°	2
70	CTOL		2
95	VTOL	0°	2
70	CTOL		2
95	VTOL	20°	2

Power Setting (%)	Mode	β_V	Time (Minutes)
70	CTOL	20°	2
95	VTOL	30°	2
70	CTOL		6
95	VTOL	40°	2
70	CTOL		2
100	VTOL	0°	1/2
70	CTOL		2
100	VTOL	20°	1
70	CTOL		2
100	VTOL	30°	1/2
70	CTOL		2
100	VTOL	40°	1
70	CTOL		2

During dual engine operation at idle in VTOL and $\beta_V = 0°$, a fire occurred in the cross-over duct insulation just below the diverter valves. Both engines experienced over-temperature condition during this run and consequently were pulled and inspected.

Due to this condition, the engines will not be run under 70% rpm at any time in the VTOL mode.

As β_V was increased during subsequent VTOL runs, high temperatures were experienced in the main wheel well area and landing gear struts. The main landing gear was then wrapped with insulation and the wheel well area was enclosed. No further over-temperatures occurred in these areas.

Some pitch fan scroll leakage was evidenced, and a seal and baffle strip were installed on the fan scroll as a fix. Additional insulation was also installed in the pitch fan area.

As β_V was increased in the VTOL mode, the thrust reverser doors and inlet louvers developed an oscillation. The louver linkage was modified and a damper was installed on the exit doors as a result.

Access doors were added in the engine cover and canoe for fire protection and inspection purposes.

2.6.9 Thrust Spoiler Test - Dual Engine

The thrust spoiler test was run in CTOL mode with both engines as follows:

Power Setting (rpm)	Spoiler Position	Time
90%	12 1/2%	30 Sec
↕	25%	↕
	37 1/2%	
	50%	
90%	62 1/2%	30 Sec
70%	Retracted	2 Min
95%	12 1/2%	30 Sec
↕	25%	↕
	37 1/2%	
	50%	
95%	62 1/2%	30 Sec
70%	Retracted	2 Min
100%	25%	30 Sec
↕	50%	↕
	75%	
100%	100%	30 Sec

At the conclusion of the 95% rpm runs, the titanium aft center fairing was replaced with a heavier gage material, due to cracking of the original installation.

Exhaust gases entered the fuselage tail pipe opening around the shrouds causing the structure to reach temperature limits. Stainless steel finger seals were installed in this opening which resolved the temperature problem.

The fuselage Fiberglas insulation also frayed out just aft of the tail pipe opening during 100% rpm and 50% spoiler. This was replaced by .016" stainless steel over Min-K insulation.

2.6.10 Single Engine Diverter Valve Tests

This test was successfully run as follows:

Mode	L/H Eng. rpm	R/H Eng. rpm
CTOL	90%	70%
VTOL	90%	70%
CTOL	95%	70%
VTOL	95%	70%
CTOL	100%	70%
VTOL	100%	70%
CTOL	70%	100%
VTOL	70%	100%

2.6.11 Pitch Fan Thrust Reverser Test

With both engines at 100% rpm in VTOL mode and $\beta_V = 0°$, the control stick was moved rapidly from extreme forward and aft positions to center and from half forward and aft positions to stop and center. (Both directions from each position.) Some pitch fan exit door flutter was experienced while at the aft stick positions.

Dampers were installed on both pitch fan exit doors and were tested successfully on XV-5A aircraft S/N 24505.

2.6.12 Flight Mode Conversion

The automatic trim switches were de-activated during this test due to design group's request. Simulator tests indicated that these switches are unnecessary, and were deleted.

Conversions were made from CTOL to VTOL and from VTOL to CTOL at 70% and 90% engine rpm. An abort from CTOL to VTOL and from VTOL to CTOL was also accomplished at 90% engine rpm.

2.6.13 Fan Overspeed Cutback

Performance of the Fan Overspeed Cutback system was satisfactory in all respects. With engine run experience obtained after the noted modifications, and further information obtained on the XV-5A S/N 24505, it is felt that the XV-5A can be flown with the minimum of risk due to temperature problems. It should be noted that the heating compliance tests are not entirely applicable to clearing the aircraft in forward flight. The XV-5A is adequately instrumented to determine powerplant operational adequacies with minimum risk to the aircraft during forward flight.

2.7 ENGINE RUN ELECTRICAL SYSTEMS CHECKOUT

The electrical system checkout performed satisfactorily as outlined in the referenced Test Procedure with the exception of the battery test. The battery test will be performed later just before first flight.

2.8 AUTO-STABILIZATION TESTS

Auto-stabilization tests as specified in Report No. 136 were completed satisfactorily.

In addition to the tests specified, frequency response tests were made on the wing fan exit louver actuators and pitch control door actuator for roll, yaw and pitch inputs to the amplifier.

During checkout, an instability of the pitch system was observed for gain settings above "4" on the pilot's gain control panel. The frequency of oscillation was determined to be 14 cps and was the result of body bending disturbing the gyro package in response to pitch door motion. A filter network was subsequently installed in the amplifier to eliminate the oscillation.

2.9 FAN FLIGHT TRIM RATES

The tests were performed on aircraft S/N 24506 in accordance with the referenced Test Procedure Section of Report No. 136 with the following exceptions: Horizontal stabilizer rate produced by longitudinal stick position was not obtained because this control function was discontinued; the aileron and rudder trim tab rates were not obtained on XV-5A aircraft S/N 24505.

In addition, the emergency trim system was tested and was found to be functioning properly.

The following are the trim rates found:

1. Pitch Control Door $2.39^\circ/s$, $\beta v = 0$

2. Horizontal Stab. $3.98^\circ/s$, $\beta v = 30$

3. Horizontal Stab. $1.15^\circ/s$, CTOL

4. Horizontal Stab. during conversion $6.51^\circ/s$, #1 Hyd. System

5. Horizontal Stab. during conversion 7.52° /s, #2 Hyd. System

6. Horizontal Stab. during conversion 6.42° /s, Both Systems

7. Horizontal Stab. during conversion 7.99° /s, Engines On

8. Roll Trim LH Wing .652 Deg β_s/Sec

9. Roll Trim RH Wing .719 Deg β_s/Sec

10. Yaw Trim LH Wing .949 Deg β_v/Sec

11. Yaw Trim RH Wing .855 Deg β_v/Sec

12. Trim Transfer Point β_v = 16 Degrees

13. Thrust Vector Actuator 3.79 Deg β_v/Sec

2.10 LANDING GEAR TESTS

The landing gear functional tests were conducted to demonstrate the sequencing and operation of the landing gear in both the normal and emergency systems. These tests were performed on aircraft S/N 24505 and will be repeated on aircraft S/N 24506 prior to flight.

2.10.1 Brake Check

Brake checks were performed per vendor specifications and brakes were found to be properly installed and adjusted.

2.10.2 CTOL Mode Landing Gear Functional Test

The aircraft was prepared for test by being placed in the CTOL flight mode, and with the landing gear in the down (forward) position. The gear was cycled through the use of the landing gear selector switch in the cockpit. Retraction and subsequent extension of the gear was normal in all respects. Cockpit position indicators and warning horn and light operation were normal.

2.10.3 VTOL Mode Main Landing Gear Functional Test

The aircraft was placed in the VTOL flight mode with the landing gear in the down (aft) position. The gear was cycled through the use of the landing gear selector switch in the cockpit. Retraction and extension of the

landing gear was normal. Cockpit position indicators and warning horn and light operation were normal.

2.10.4 Downlock Over-ride Check

The main landing gear wheels were placed on blocks which allowed the downlock Microswitch to open and require use of the over-ride button to bring the landing gear up. Each wheel was tested individually and then together. In each case the over-ride button had to be used to select the UP position on the switch. Retraction from that period was normal.

2.10.5 STOL Over-ride

The aircraft was placed in the VTOL flight mode with the landing gear down. The STOL over-ride switch was placed in the "STOL" position, and the landing gear moved to the CTOL position. Landing gear position indicators operated normally and the STOL warning light illuminated. Returning the over-ride switch to "Normal" returned the landing gear to the VTOL position. Indicators and light returned to the normal state.

2.10.6 Landing Gear Retraction Under Load Factor

The main landing gear was cycled while under a simulated inertia load of 1-1/2 g's. Lead shot bags were placed along the landing gear struts so that the simulated load acted at the original landing gear center of gravity. No discernible changes in gear retraction or extension were noted.

2.10.7 Emergency Pneumatic Extension System

The landing gear was extended several times under various pneumatic pressures.

No load extensions were made along with minimum pressure and maximum airload extensions. The tests showed the requirement for increasing the diameter of the restrictor orifice in the emergency system from the original .0105 inch diameter to .0156 inch diameter. This change resulted in much better extension rates, and more positive stops at the travel limit.

Extensions using 1700 psig nitrogen pressure and with the main landing gear under a simulated drag load of 170 pounds per side were made with times and rates well within the established design limits. The nitrogen system low pressure warning light illuminated at 1675 psig.

2.10.8 VTOL - CTOL Mode Change - Aircraft Resting on Its Own Wheels

This particular portion of the functional test has not been accomplished to date. This has come about as a result of landing gear and landing gear door modifications made during engine runs. The mode change test will be accomplished, however, prior to flight-required mode change.

2.11 CONTROLS PROOF LOADS

The controls proof loads test was run as outlined in the following paragraphs. All loads were applied by hydraulic cylinders with the exception of the lateral stick and collective control stick. These loads were applied by hand using a spring scale or by dead weight.

2.11.1 Rudder - CTOL

The rudder was restrained in the neutral position and a 300-pound load was applied to each rudder pedal individually.

2.11.2 Elevator - CTOL

The elevator was restrained in the neutral position and a 200-pound load was applied forward and aft to the control stick grip.

2.11.3 Aileron - CTOL

The aileron tabs were restrained in the neutral position and a 100-pound load was applied laterally to the control stick grip in both directions.

2.11.4 Elevator Cable Stretch - CTOL

In VTOL mode, the control stick was restrained in the full aft position by a 150-pound force. With the horizontal stabilizer in the full L.E. down position, an increasing down load was applied to the elevators until the control stick just cleared the aft stop. The elevator moved down 6 degrees before first movement of the stick occurred.

2.11.5 Throttle Test

A 75-pound aft load was applied to both throttles (separately) with the load reacted by bottoming of the lower throttle mechanism.

2.11.6 Collective Control Stick

The collective control stick was loaded to 150-pounds in both the up and down directions. Loads were reacted by bottoming of the stick mechanism in the cockpit.

2.11.7 Control Stick and Rudder Pedals - VTOL

In the VTOL mode, the control stick and rudder pedals were displaced to their extreme positions (separately) with hydraulic pressure on and held firmly as hydraulic power was shut off. A spring scale was used to pull the stick or rudder pedal in the opposite direction to which it was originally displaced, and the force to bring it to the opposite cockpit stop was recorded.

During lateral stick loading, the control stick pivot tube pulled out of its aft bearing support. A fix was made and the load test was successfully completed.

2.12 WEIGHTS - BALANCE AND FUEL TESTS

The weight and balance tests on aircraft S/N 24505 were run to determine total aircraft weight and the variation in center of gravity position with various fuel quantities.

The aircraft landing gear was placed in the VTOL (aft) position and all installed equipment was in place or simulated. The aircraft was placed on standard aircraft scales and leveled for the empty weight measurements.

Forward fuel tank quantity and center of gravity measurements were run as installed in the aircraft. The aft main and dorsal tanks were calibrated out of the aircraft in special holding fixtures. Fuel centers of gravity were obtained on all tanks for the following pitch attitudes:

Level, ±5, ±10, and ±15 degrees.

2.12.1 Empty Weight - Aircraft Level

The aircraft was positioned as noted in Paragraph 2.12 above. A survey of installed equipment showed the following items to be missing:

1. Fiberglas flap hinge fairings
2. Throttle quadrant cover

3. Annunciator panel

4. Signal conditioner box

5. Telemetry box

6. Seat mode speed sensor

7. Engine compartment cooling fans and gear boxes (20-pounds of lead shot were added at each fan and gear box location to simulate component weights)

8. Heat shield modification for main landing gear

9. Chine rail installation above tailpipe exits and modification to original Min-K insulation installation in the tailpipe area.

10. Pilot seat rocket motor.

The net weight, not including the above items, was 7,874-pounds. The center of gravity at this weight was located at F.S. 245.52, W.L. 117.48. Omitted aircraft components will be compensated for using the IBM weight and balance computer program.

2.12.2 <u>Fuel Quantities - Level Attitude</u>

The following fuel quantities were required to fill the tanks indicated:

Forward Fuselage Tank:	1600 pounds
Aft Main Tank:	830 pounds*
Dorsal Tank:	797 pounds

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* Slightly more fuel may be placed in this tank when installed in the aircraft as it will be filled through the dorsal tank and may be filled completely. During testing the tank was filled through its own filler cap, which will not allow the tank to become completely filled.